UNCLASSIFIED

AD 414585

DEFENSE DOCUMENTATION CENTER

FOR

SCIENTIFIC AND TECHNICAL INFORMATION

CAMERON STATION, ALEXANDRIA, VIRGINIA

UNCLASSIFIED

63-4-5

414585

LIQUID ROCKET PLANT

COPY NO. 15 August 1963

WEAPON SYSTEM 107A-2
PRODUCT ENGINEERING PROGRAM

MONTHLY PROGRESS REPORT

Report 212/SA3-2.2-M-2 Contract AF 04(694)-212/SA3

AEROJET GENERAL TIRE GENERAL

AEROJET-GENERAL CORPORATION
SACRAMENTO, CALIFORNIA

5610

WEAPON SYSTEM 107A-2

PRODUCT ENGINEERING PROGRAM

Contract AF 04(694)-212/SA 3

1 July through 31 July 1963

Prepared by

AEROJET-GENERAL CORPORATION
Liquid Rocket Plant
Sacramento ?, California

Prepared for

BALLISTIC SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
Norton Air Force Base, California

5310

AEROJET GENERAL LIQUID ROCKET PLANT SACRAMENTO, CALIFORNIA

A SUBSIDIARY OF THE GENERAL TIRE & RUBBER COMPANY

FOREWORD

This report is the second in a series of monthly reports prepared in accordance with AFBM Exhibit 58-1 and submitted in partial fulfillment of Contract AF 04(694)-212, Supplemental Agreement No. 3.

Direction for contract performance is provided by C. L. D'Ooge, Program Manager, Research and Advanced Technology Division, Liquid Rocket Plant.

The contract for the continuation of the Product Engineering Program is made up of four projects:

	PROJECT TITLE	PROJECT ENGINEER
I.	Coated Metallic Thrust Chambers	D. G. Harrington
II.	Expandable Nozzle	D. M. Green
III.	Combustion Instability Scaling Concepts	F. H. Reardon
IV.	Ablative Thrust Chamber Feasibility	T. A. Hughes

TABLE OF CONTENTS

TABLE LIST

FIGURE LIST

I. COATED METALLIC THRUST CHAMBERS

A. INTRODUCTION

1. Purpose

The objective of this project is to develop a reliable thermal barrier capable of surface temperature operation above 3300°F for use on regeneratively cooled thrust chambers employing N_2O_4/AeroZINE 50 propellants. Thermal barrier coatings are used for improving liquid rocket engine performance through reductions in film-cooling flow rate requirements.

2. Approaches

The development of the thermal barriers will be accomplished through:

a. Full-scale testing of coated YLR91-AJ-5 engines (Titan II second stage) utilizing fuel-film-cooling flow rates of 5% or less.

b. Laboratory thermal shock, thermal conductivity, erosion-corrosion, and metallographic testing.

B. PROGRESS DURING REPORT PERIOD

1. Full Scale Testing

The first coated combustion chamber, S/N 111, was fired once on 16 July 1963 and once on 18 July 1963. The duration of each test was 12 seconds, as scheduled in the program plan, and 17.6% fuel-film-cooling was employed.

There was no visible damage to the coating in the throat area after the first test, but there was flaking of the outside layer of zirconia forward of the throat in the convergent section. The flaking extended along

I. COATED METALLIC THRUST CHAMBERS

A. INTRODUCTION

1. Purpose

The objective of this project is to develop a reliable thermal barrier capable of surface temperature operation above 3300°F for use on regeneratively cooled thrust chambers employing N_2O_4/AeroZINE 50 propellants. Thermal barrier coatings are used for improving liquid rocket engine performance through reductions in film-cooling flow rate requirements.

2. Approaches

The development of the thermal barriers will be accomplished through:

a. Full-scale testing of coated YLR91-AJ-5 engines (Titan II second stage) utilizing fuel-film-cooling flow rates of 5% or less.

b. Laboratory thermal shock, thermal conductivity, erosion-corrosion, and metallographic testing.

B. PROGRESS DURING REPORT PERIOD

1. Full Scale Testing

The first coated combustion chamber, S/N 111, was fired once on 16 July 1963 and once on 18 July 1963. The duration of each test was 12 seconds, as scheduled in the program plan, and 17.6% fuel-film-cooling was employed.

There was no visible damage to the coating in the throat area after the first test, but there was flaking of the outside layer of zirconia forward of the throat in the convergent section. The flaking extended along

the crowns of the tubes 3 to 8 in. forward of the throat, 360° around the wall of the combustion chamber (Figure 1). The increased amount of flaking in the region from 5 to 8 in. forward of the throat was possibly the result of this area corresponding to the location of a throat plug that is inserted for both pre- and post-fire leak-checking at 50 psi, as shown in Figure 2. Close examination after the test indicated that none of the inner three layers had flaked or eroded, except at the injector flange where separation of the entire coating from the substrate had started in three areas (Figure 3). The coating on this flange was approximately three times thicker than on the rest of the combustion chamber.

After the second firing, the flaking of the zirconia top layer had extended an additional three inches downstream and reached the minimum diameter of the throat (Figure 4). A few of the tubes had little or no flaking in the throat and none of the under layers appeared to have been removed (Figure 5). One six-inch long area of the thick coating on the flange was lost at the substrate surface (Figure 6).

Flaking of the top layer of the coating was a persistent problem in laboratory thermal shock testing on the previous contract (AF 04(647)-652/5A4) when multiple tube specimens were used. However, the coating used on the first chamber, S/N 111, was chosen on the basis of laboratory tests using single-tube specimens. In these tests the best coating was a gradated Nichrome-zirconia type, and no damage occurred in 70 thermal cycles for Specimen No.'s 83, 84, 85 and 86.* After the chamber had been

* BSD-TDR-119, "Development of Thermal Barrier Coatings for Regeneratively-Cooled Rocket Engine Thrust Chambers," 28 June 1963, pp 21-23; Tables 6, 12, 14; Figure 27.

coated, a final proof-check of the coating applied to the chamber was made by preparing multiple-tube thermal shock specimens and employing identical surface preparation and spraying conditions as used on the chamber. The result was failure in two cycles in each of four tests on Specimen No.'s 105 and 106.** Since the coating in the throat area of the chamber S/N 111 failed in an identical manner in two cycles, the validity of the laboratory tests is verified, and only multiple-tube specimens will be utilized in subsequent laboratory work.

2. Design

The design of the injector for obtaining 5% fuel-film-cooling was completed. The 17th channel ring of a 2S1N-0 injector, which has two rows of holes for 17.6% fuel-film-cooling, will be replaced by a ring with one row of 148 holes to obtain 5% fuel-film-cooling. The holes will be drilled at an angle such that the fuel will impinge at the flange seal. Two additional coolant thermocouple taps will be provided so that three coolant exit temperatures can be measured on each test.

The design for the chamber coating fixture is being modified to include a dust collection system, boot covers on the polished shafts, a more powerful motor, increased runout, swivel coasters, a chamber flange guard ring, and a more flexible grit blasting hose. This design is 90% complete.

** Ibid, p 30; Tables 11 and 16; Figure 27

3. Fabrication

A fabrication order for the 5% fuel-film-cooling injector was prepared. Completion by 7 August 1963 is expected.

Fabrication of the second YLR91-AJ-5 chamber, S/N 112, which was partially completed on the previous contract, has been resumed and is scheduled for completion 14 August 1963.

Fabrication of the third and fourth combustion chambers has been started and deliveries on schedule are anticipated.

It is now expected that at least one of the combustion chambers will be re-useable after testing by removing the coating by grit blasting and re-coating. Therefore, fabrication of the fifth combustion chamber may not be required.

The full-scale coating fixture is scheduled to be modified during the month of September after the second chamber has been coated. All critical parts have been ordered. A few improvements will be made before coating the second chamber. Since most of the planned changes are for convenience of operation and protection from wear, the quality of the coating on the second chamber will not be jeopardized by not having these modifications completed.

[illegible]. NEXT REPORT PERIOD

The first coated combustion chamber, S/N 111, will be test fired for five sec after the injector is fabricated and the flange cover re-coated. The thermal resistance of the coating remaining on the tubes is theoretically sufficient to prevent tube burnout with 5% film cooling. Also, laboratory

results on the previous program showed no further damage to identical coatings after as many as 29 additional cycles after part of the zirconia overlay had flaked off.

Laboratory tests will be employed to determine what coating should be applied to the second chamber, S/N 112. The second chamber will then be coated.

A quality control contract will be initiated with a vendor for thickness measuring methods.

TABLE I-1

COATED METALLIC THRUST CHAMBER PROGRAM [illegible]

COATED METALLIC THRUST CHAMBER	PRESENT SCHEDULE DAY	MO	YR
PHASE I, ANALYSIS AND DATA EVALUATION			
A. Heat Transfer			
1. Initiated			
2. Completed			[illegible]
B. Thermal Stress			
1. Initiated			3
2. Completed		[illegible]	[illegible]
C. Erosion and Corrosion			
1. Initiated			
2. Completed	[illegible]		[illegible]
D. Data Evaluation			
1. Initiated	1		[illegible]
2. Coating Selected (Chamber [illegible])	[illegible]		
3. Coating Selected (Chamber [illegible])		[illegible]	[illegible]
4. Coating Selected (Chamber [illegible])			
5. Coating Selected (Chamber [illegible])	[illegible]	[illegible]	
6. Completed	[illegible]		
E. Specification for Best Thermal Barrier			
1. Initiated	[illegible]		4
2. Completed	[illegible]		4
PHASE II, LABORATORY INVESTIGATIONS			
A. Laboratory Investigation Initiated	1	[illegible]	
1. Thermal Shock Tests		[illegible]	[illegible]
2. Thermal Conductivity Tests	1	[illegible]	3
3. Erosion-Corrosion Tests	1	[illegible]	[illegible]3
4. Coating Application Technique Developed	1	6	63
5. Metallography	1	6	63
B. Laboratory Investigation Completed	31	5	64

TABLE I-1 (Cont.)

CURRENT CHANGE	COATED METALLIC THRUST CHAMBERS	PRESENT SCHEDULE DAY	MO	YR	PRIOR	CY 63 J	J	A	S	O	N	D	CY 64 J	F	M	A	M	
	PHASE III, QUALITY CONTROL																	
	A. Contract for Studies Initiated	15	[illegible]	63	◇			▲										
	1. Raw Material Control	15	[illegible]	63	◇			▲										
	2. Process Control	[illegible]		63	◇			▲										
	3. Destructive Sampling	15	8	63	◇			▲										
	4. Nondestructive Testing	15	[illegible]	63	◇			▲										
	B. Contract for Studies Completed	27	12	63								▲						
	C. Laboratory Proof Test																	
	1. Initiated	15	[illegible]	63				▲										
	2. Completed	1	12	63								▲						
	PHASE IV, FULL SCALE TESTING																	
	A. Thrust Chamber 1																	
	1. Ready for Buildup	1	6	63		▲												
	2. Ready for Testing	7	7	63			▲											
	3. Testing Completed	20	[illegible]	63				▲										
	B. Thrust Chamber 2																	
	1. Release for Fabrication	[illegible]	[illegible]	63														
	2. Ready for Coating	1		[illegible]				▲										
	3. Ready for Buildup	15		[illegible]				▲										
	4. Ready for Testing	12	[illegible]	[illegible]					▲									
	5. Testing Completed	[illegible]		[illegible]						▲								
	C. Thrust Chamber 3																	
	1. Release for Fabrication	[illegible]	[illegible]	[illegible]														
	2. Ready for Coating	[illegible]	[illegible]	[illegible]						▲								
	3. Ready for Buildup	3	[illegible]	[illegible]						▲								
	4. Ready for Testing	1	11	[illegible]							▲							
	5. Testing Completed		[illegible]	[illegible]								▲						
	D. Thrust Chamber 4																	
	1. Release for Fabrication	[illegible]	[illegible]	[illegible]														
	2. Ready for Coating	[illegible]	[illegible]	[illegible]								▲						
	3. Ready for Buildup		1	64									▲					
	4. Ready for Testing	[illegible]	[illegible]	64										▲				
	5. Testing Completed	[illegible]	[illegible]	64											▲			
	E. Thrust Chamber 5																	
	1. Release for Fabrication	1	[illegible]	[illegible]					▲									
	2. Ready for Coating	17	[illegible]	64											▲			
	3. Ready for Buildup		[illegible]	64												▲		
	4. Ready for Testing	[illegible]	4	64												▲		
	5. Testing Completed	3[illegible]	[illegible]	64													▲	
	F. Injector 1																	
	1. Release for Modification	[illegible]	7	63			▲											
	2. Ready for TCA Buildup	1	[illegible]	[illegible]				▲										

COATING DAMAGE AFTER TEST 1 OF CHAMBER 1

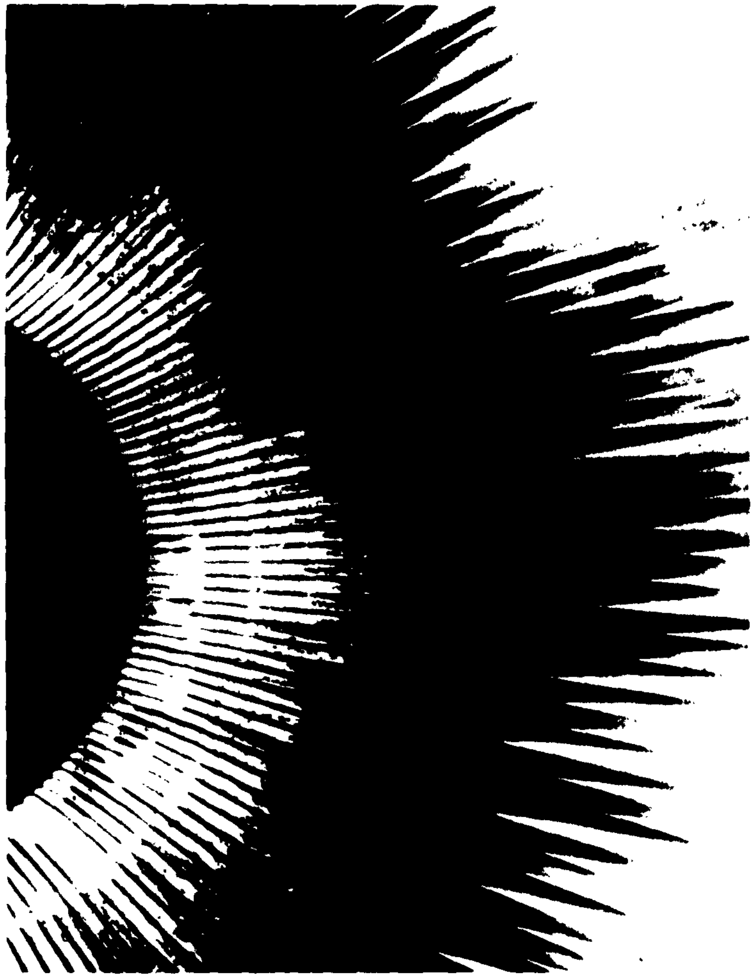
Figure IV-1

THROAT PLUG USED FOR PRE-AND POST-FIRE LEAK CHECKING

Figure IV-2

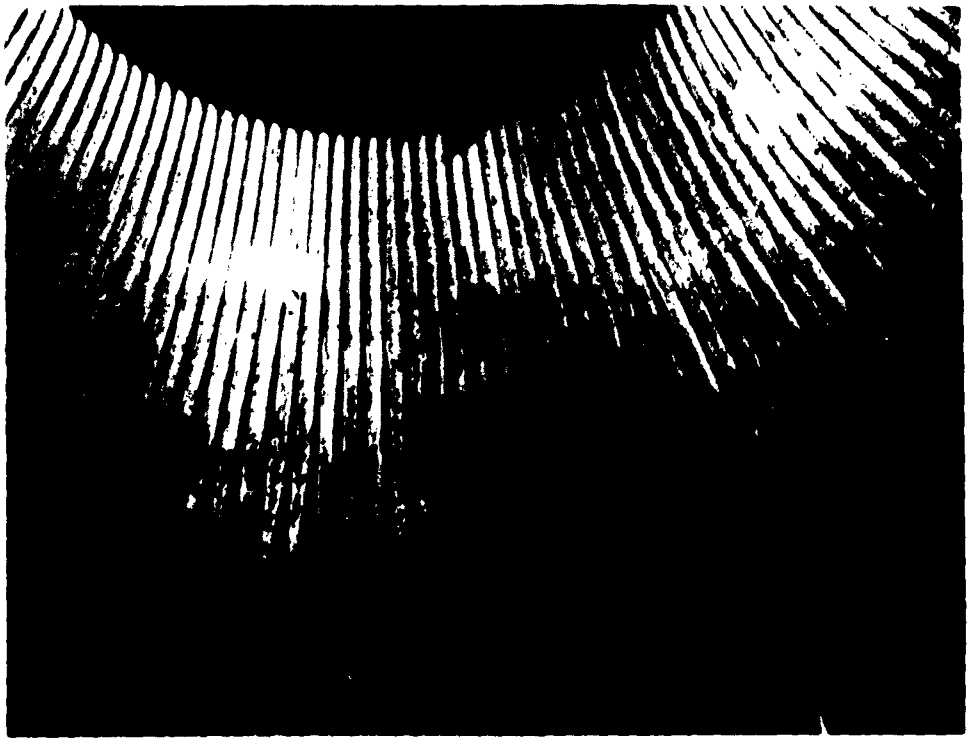
BEGINNING OF SEPARATION OF COATING FROM
FLANGE COVER AFTER TEST 1 OF CHAMBER 1

Figure IV-3



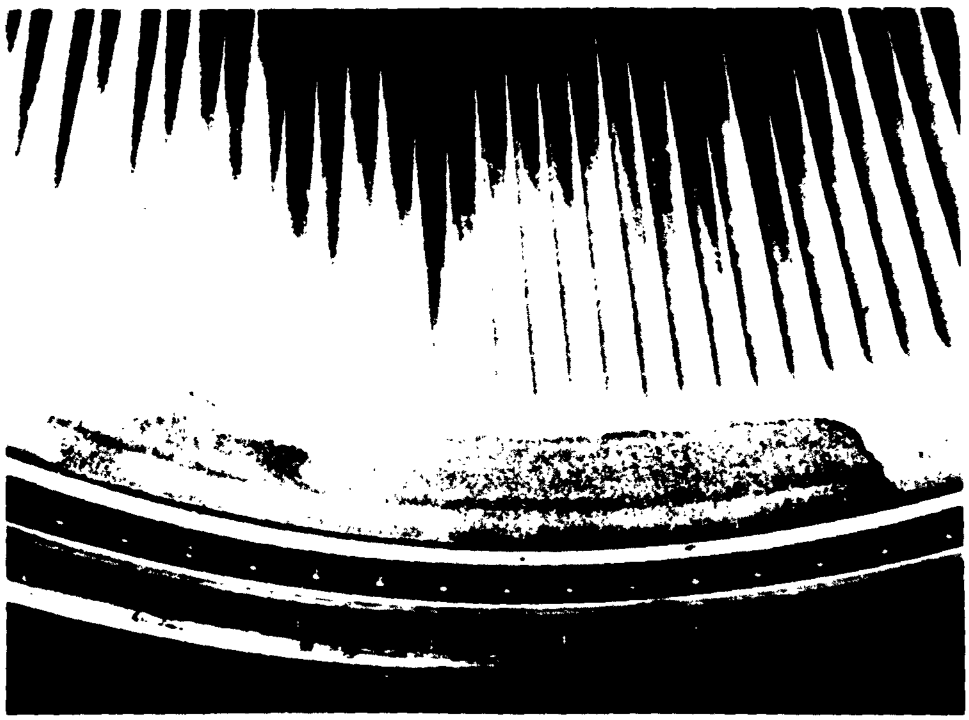
COATING DAMAGE AFTER TEST 2 OF CHAMBER 1

Figure IV-4



COATING DAMAGE AFTER TEST 2 OF CHAMBER 1, THROAT AREA CLOSE-UP

Figure IV-5



LOSS OF COATING FROM FLANGE COVER AFTER TEST 2 OF CHAMBER 1

Figure IV-6

II. EXPANDABLE NOZZLES

A. INTRODUCTION

1. Purpose

The purpose of this project is to develop and test efficient packaging of high expansion ratio nozzle exit cones for rocket propulsion systems.

2. Approach

The program objectives will be accomplished by a three-phase (design, fabrication and experimental) effort. Work will be directed to simulate the operational nozzle conditions of the Titan II second-stage engine with both metallic and non-metallic nozzle expansion cone skirt extensions. Thrust-vectoring and internal gas dynamic conditions will be investigated utilizing a bipropellant liquid rocket engine (12,000-lb thrust) mounted on a multi-axis thrust stand. A series of flight simulation tests in a wind tunnel will be conducted with scale models using various forebody shapes.

B. PROGRESS DURING REPORT PERIOD

1. Design

Preliminary corrections have been made to the theoretical heat transfer program on the basis of test firings (Lark and full-scale engine with modified injector) conducted on the previous contract*.

Design modifications to the Lark engine have been initiated, and Lark-size, non-metallic pressure-deployment nozzles are being designed.

---

* AF 04(647)-652, SA 33

Battleship nozzle designs have been initiated for thrust vector control and internal gas dynamics testing on the 12,000-lb thrust engine.

Design drawings for engine assembly and installation have been completed for the large scale test program at the Arnold Engineering and Development Center. Metallic and non-metallic expandable nozzle designs have been started.

Preliminary design data have been compiled for the flight simulation test program.

2. Fabrication

A laboratory model of the gas-pressure-tube deployment system was completed and tested. Although the fabrication of this model was somewhat crude, it did demonstrate the feasibility of such a system to reduce flutter in rubber nozzles at expansion.

3. Testing

Lark material testing has been initiated, but results are not yet available.

Test stand build-up is progressing at the Azusa proving grounds for the thrust vector control program.

For the flight simulation program, arrangements have been made to use the 21-in. hypersonic wind tunnel at the Jet Propulsion Laboratory.

C. NEXT REPORT PERIOD

Nozzle designs will be completed for the Lark, 12K engine, full-scale engine and wind tunnel models. The test stand for the thrust vector control program will be completed and checked out. A pre-test conference will be held at the Arnold Engineering Development Center.

TABLE II [illegible]

EXPANDABLE NOZZLE MILEPOST CHART

Current Change	Expandable Nozzles	Present Schedule Day	Mo	Yr	Prior	[illegible] J	J	A	S	O	N	D	[illegible] J	F	M	A	M	
	PHASE I, ENGINEERING DESIGN																	
	A. Refinement and Correlation																	
	[illegible]																	
	1. Initiated	1	[illegible]	[illegible]		◆												
	2. Completed	31	5	64														◆
	B. [illegible] Hardware																	
	1. Initiated	1	6	63		◆												
	2. Completed	31		63					◆									
	C. [illegible] Hardware																	
	1. Initiated	[illegible]	[illegible]	[illegible]		◆												
	[illegible]	[illegible]		[illegible]						◆								
	D. [illegible] Hardware																	
	1. [illegible]	[illegible]	[illegible]	[illegible]			◆											
	[illegible]											◆						
	[illegible]																	
	[illegible]																	
	[illegible]							◆										
	[illegible]														◆			
	PHASE II, [illegible]																	
	A. [illegible]																	
	[illegible]						◆											
	[illegible]		[illegible]										◆					
	[illegible]																	
	[illegible]						◆											
	[illegible]												◆					
	[illegible]																	
	[illegible]								◆									
	[illegible]														◆			
	[illegible]																	
	[illegible]								◆									
	[illegible]														◆			
	[illegible]																	
	[illegible]										◆							
	[illegible]															◆		
	PHASE III, [illegible]																	
	[illegible]																	
	[illegible]								◇									
	[illegible]														◆			
	[illegible] ([illegible])																	
	[illegible]								◆									
	[illegible]														◆			
	[illegible] ([illegible])																	
	[illegible]												◆					
	[illegible]															◆		
	[illegible]																	
	[illegible]																	
	[illegible]											◆						
	[illegible]			[illegible]											◆			
	[illegible]																	
	[illegible] ([illegible])																	
	[illegible]	[illegible]	[illegible]	[illegible]									◆					
	[illegible]			[illegible]														◆

III. COMBUSTION INSTABILITY SCALING CONCEPTS

A. INTRODUCTION

1. Purpose

The combustion instability scaling program is an effort to preduct analytically the longitudinal modes of instability of a subscale rocket motor, to determine from testing of this motor the longitudinal mode stability limits; to compare these results; and finally to utilize the results for determining the occurrence of transverse modes of instability for larger rocket motors.

2. Approaches

This program is concerned with the stability of rocket engines as it is affected by chamber length, chamber pressure, and propellant mixture ratio. An empirical analysis will be attempted to determine the axial combustion distribution of this engine, which is required by the theoretical analysis.

For each test, a chamber pressure and a mixture ratio will be specified and held constant throughout the test. Once the thrust chamber has reached steady-state operating conditions, the chamber length will be increased during the test to study the stability effects resulting from the variation in chamber length. Two firings will be required for each operating point to keep the duration approximately 5.0 sec for each test. For the next test, the chamber pressure and/or the mixture ratio will be changed, the motor will be returned to its initial chamber length (a length to diameter ratio approximately equal to unity), and the test series continued.

B. PROGRESS DURING REPORT PERIOD

1. Thrust Chamber Assembly Design

a. Combustion Chamber

Figure III-1 is an isometric drawing of the 10K, variable length thrust chamber assembly. It illustrates the various components and their location.

The injector, hydraulic cylinders, and the four recirculating ball bearing assemblies are externally supported. The combustion chamber is restrained by the two hydraulic cylinders which are secured to the combustion chamber at the nozzle flange. The splines and the recirculating ball bearing supports prevent rotation.

The chamber will have the capability of increasing its length by sixteen inches from a minimum chamber length of six inches. After the motor is ignited, it will be allowed to reach steady-state operating conditions (in approximately two sec) at a chamber length of six in. An electronic timer will then open a pilot valve on the vent side of the hydraulic cylinders, allowing the hydraulic pressure on the relief side to decrease. The rate of pressure decay will be regulated by orifices in the vent system. The pressure side of the piston will be maintained at a pressure of about 2000 psig, which will insure a uniform increase in chamber length even if the motor becomes unstable and the chamber pressure begins to oscillate.

A water-cooled "Aerofoil" will be introduced near the chamber exit plane. This foil will measure nine temperatures and eight stagnation pressures at its leading edge and eight static pressures on the bottom. It

will have the capability of being mounted at various distances from the injector face within the final six in. of the chamber to allow more effective investigations of combustion distribution.

Since the Aerofoil will be removed for the instability portion of the program, it is presently planned to measure static pressures along the chamber wall in six locations in the final six in. of the chamber, or one static pressure point every inch. Two high-frequency transducers will also be incorporated in this test series, one mounted directly in the center of the injector with the second mounted in the chamber wall at the nozzle flange. These transducers will furnish the acoustical information at the boundaries of the chamber necessary for the primary objectives of the program.

Figure III-2 is a preliminary illustration of the thrust frame for the variable length thrust chamber assembly. The thrust chamber assembly will be assembled in the thrust frame before it is delivered to the test area. Since the propellant valves will also be mounted to the thrust frame, the time required for test preparation in the test area is expected to be minimum.

b. Nozzle

Figure III-1 illustrates the original nozzle configuration, a flat plate across the exit of the chamber with six equally sized, equally spaced orifices whose sum area was equivalent to the area of a conventional one-element nozzle. Further design has revealed that this configuration is not feasible. Figure III-3 illustrates the latest nozzle. The total area of all the orifices is 11.66 in.^{2}. Instead of equally sized holes, there

are two orifice sizes, with the center orifice accounting for one-fourth of the total area.

The thermal and erosion protection of the nozzle plate will be accomplished by either one or all of the following methods: protective coating of the hot gas side of the plate, water-cooling, and use of graphite inserts in the convergent section of the orifices. The critical design considerations of the nozzle plate were the length of the convergent sections and the multiple orifice concept. According to the sensitive combustion time lag theory, the nozzle geometry exerts a definite damping effect on the longitudinal modes of instability experienced by the combustion chamber. The amount of damping is a function of the reduced frequency, ω_f, which is proportional to the ratio of the length of the exhaust nozzle to the length of the combustion chamber. Since this motor is designed to operate unstably, the design of the exhaust nozzle should incorporate the least amount of damping possible; therefore, the condition was imposed that:

$$\frac{(\omega_f)_m}{(\omega_F)_e} \leq 0.10$$

where subscript m represents the subscale motor and the subscript e represents the prototype engine to which this motor is being scaled.

In order to determine the length of the nozzle, the following expression was solved for the nozzle length using the above condition.

$$\frac{(\omega_f)_m}{(\omega_f)_e} = 0.10 = \frac{\left(\frac{l}{L_c}\right)_m}{\left(\frac{l}{L_c}\right)_e}$$

or

$$l_m \leq 0.40$$

where l represents the length of the subsonic portion of the nozzle and L_c indicates the length of the combustion chamber. The length of the subscale chamber used in the above calculation was the minimum of six in.

The desirability of utilizing multiple orifices originates from one dimensional flow considerations. If a single orifice were used as the nozzle in this program, the flow of the hot gases from the chamber to the nozzle would result in stagnation zones and eddy currents at the intersection of the nozzle plate and the chamber wall. Also, because of streamlining, there would be significant departures from one dimensional flow. Consequently, by adopting a multiple orifice nozzle, the streamlining effects have been greatly reduced and a one-dimensional flow pattern can be more closely simulated.

Since the program is not specifically concerned with motor performance with respect to thrust and nozzle efficiency, the divergent section of the nozzle is an arbitrary design. The length of the nozzle beyond the throat area is also arbitrary, but will be of a dimension that can add structural rigidity to the nozzle plate.

As Figure III-1 illustrates, the nozzle plate is secured to the combustion chamber by bolts, facilitating replacement on the test stand in the event of damage.

c. Injector

The injector pattern will simulate the 2SIN-0 pattern which was formerly used on the YLR-91-AJ-5 second-stage engine of Titan II with the propellant combination of nitrogen tetroxide and Aerozine-50. The 2SIN-0 pattern features a like-on-like impingement with 17.62% fuel-film cooling.

The primary consideration in the determination of the subscale injector pattern was to keep the injection density as consistent as possible with that of the full-scale engine. A graphical analysis was utilized to achieve this objective.

It was also required that the detailed injection characteristics of the subscale characteristics of the subscale thrust chamber simulate those of the full-scale engine. Therefore, the propellant velocity in the feed channels, the orifice configuration, and the injection velocity correspond as nearly as possible to the full-scale injector. The model injector has six active propellant channels, alternately fuel and oxidizer. A seventh channel is used for 17.62% fuel-film coolant of the combustion chamber wall.

d. Material Considerations

At the outset of the program, a heat transfer analysis was conducted simultaneously with a preliminary stress analysis to determine the feasibility of incorporating materials in this project that are similar to those now in use with the Titan II combustion chamber and nozzle. Although the heat transfer results indicated no serious problems with this material

if the wall thickness was in the order of 0.020-in., the preliminary stress analysis indicated that this wall thickness was not satisfactory. The combination of hoop and thermal stress results in a total stress of approximately 200,000 psi, which exceeds the yield stress of the Titan II material by a factor of 7. Various materials and fabrication methods are now being studied so that a material and a wall thickness can be selected in the near future.

e. Calculation of Theoretical Instability Zones

At present, the information necessary for input to the computer program is being compiled. The modification of one of the computer programs is in process to handle this special case.

f. Test Considerations

The test stand selected for the program is C-2. The firing attitude of the stand is seventeen degrees below the horizontal, which will allow excellent camera coverage of the testing because of minimum interference of test stand superstructure and associated hardware. The test facility has more than adequate facilities for the hydraulic demands of this motor, for the combustion chamber cooling water supply, and for the special cooling water supply required by the high frequency instrumentation.

Since there exists the possibility of test stand interference and to minimize the time required for repeat testing, this subscale motor has been designed so that any discrepant component can be easily replaced while the motor is still in position on the stand. Consequently, the motor will be delivered to the test area along with five interchangeable

combustion chamber liners, three injectors, two nozzle plates, and two Aerofoils. An attempt is now in progress to maximize the accessibility of these components while the motor is in position on the stand. Because the cost of the initial installation of the motor is greater than the actual cost per test, the feasibility of conducting all of the combustion distribution tests and the instability investigations in a sixteen hour day is being considered. However, the final decision will be determined by the durability of the components in the environment of the motor calibration tests.

C. NEXT REPORT PERIOD

Final design of the combustion chamber, exhaust nozzle, and injector will begin in the next report period. Fabrication of the TCA thrust mount assembly will be initiated. The material considerations will be finalized and the wall thickness will be determined. Preliminary test requests and instrumentation requests will be submitted to the Test Division for approval. Calculation of the theoretical zones of instability will continue.

The milepost chart for this project is shown in Table III-1.

TABLE III-1

COMBUSTION INSTABILITY SCALING CONCEPTS MILEPOST CHART

CURRENT CHANGE	COMBUSTION INSTABILITY SCALING CONCEPTS	PRESENT SCHEDULE DAY	PRESENT SCHEDULE MO	PRESENT SCHEDULE YR	PRIOR	Calendar Year 1963 J	J	A	S	O	N	D	Calendar Year 1964 J	F	M	A	M	
	PHASE I, ENGINEERING DESIGN																	
	A. Concept Completed				A													
	B. Predesign																	
	1. Initiated	1	[illegible]	63														
	2. Completed	31	[illegible]	[illegible]														
	C. Final Design Initiated	1	9	[illegible]														
	1. Mount Completed	1[illegible]	9	[illegible]														
	2. Thrust Chamber Completed	1	9	[illegible]														
	3. Injector Completed	4	10	[illegible]														
	4. Actuation System Completed	11	10	[illegible]														
	PHASE II, FABRICATION INITIATED	[illegible]	[illegible]	[illegible]														
	1. Mount Completed	4	11	[illegible]														
	2. Thrust Chamber Completed	[illegible]	1[illegible]	3														
	3. Injector Completed	[illegible]	1[illegible]	[illegible]														
	4. Actuation System Completed	1[illegible]	1	[illegible]														
	5. Calibration Completed	[illegible]	1[illegible]	[illegible]														
	PHASE III, TESTING INITIATED	1[illegible]	1[illegible]	[illegible]														
	1. Test Stand Build-Up Completed	[illegible]	1	[illegible]														
	2. Calibration Firings Completed	1[illegible]	1	[illegible]														
	3. Test Program Completed	31	1	[illegible]														
	PHASE IV, ANALYSIS INITIATED	1	[illegible]	[illegible]														
	1. Calculation of Theoretical																	
	Instability Zones Completed	1[illegible]	1[illegible]	[illegible]														
	2. Completion of Comparison Between																	
	Theoretical Zones and Test Results	[illegible]	4	4														

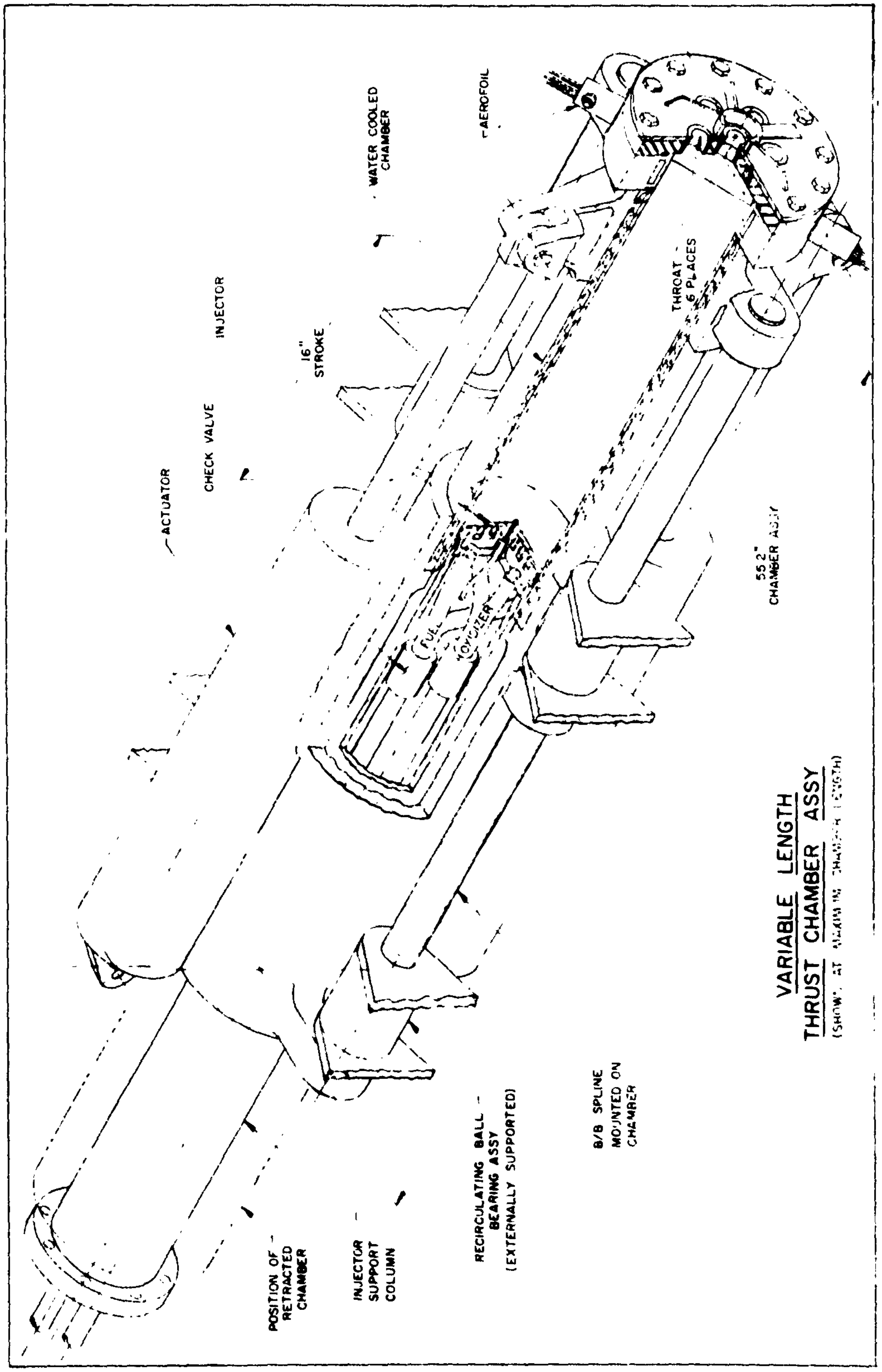

POSITION OF RETRACTED CHAMBER
INJECTOR SUPPORT COLUMN
RECIRCULATING BALL BEARING ASSY (EXTERNALLY SUPPORTED)
B/B SPLINE MOUNTED ON CHAMBER
ACTUATOR
CHECK VALVE
INJECTOR
16" STROKE
WATER COOLED CHAMBER
AEROFOIL
THROAT - 6 PLACES
FUEL
OXIDIZER
55.2"
CHAMBER
VARIABLE LENGTH
THRUST CHAMBER ASSY

Figure III-1

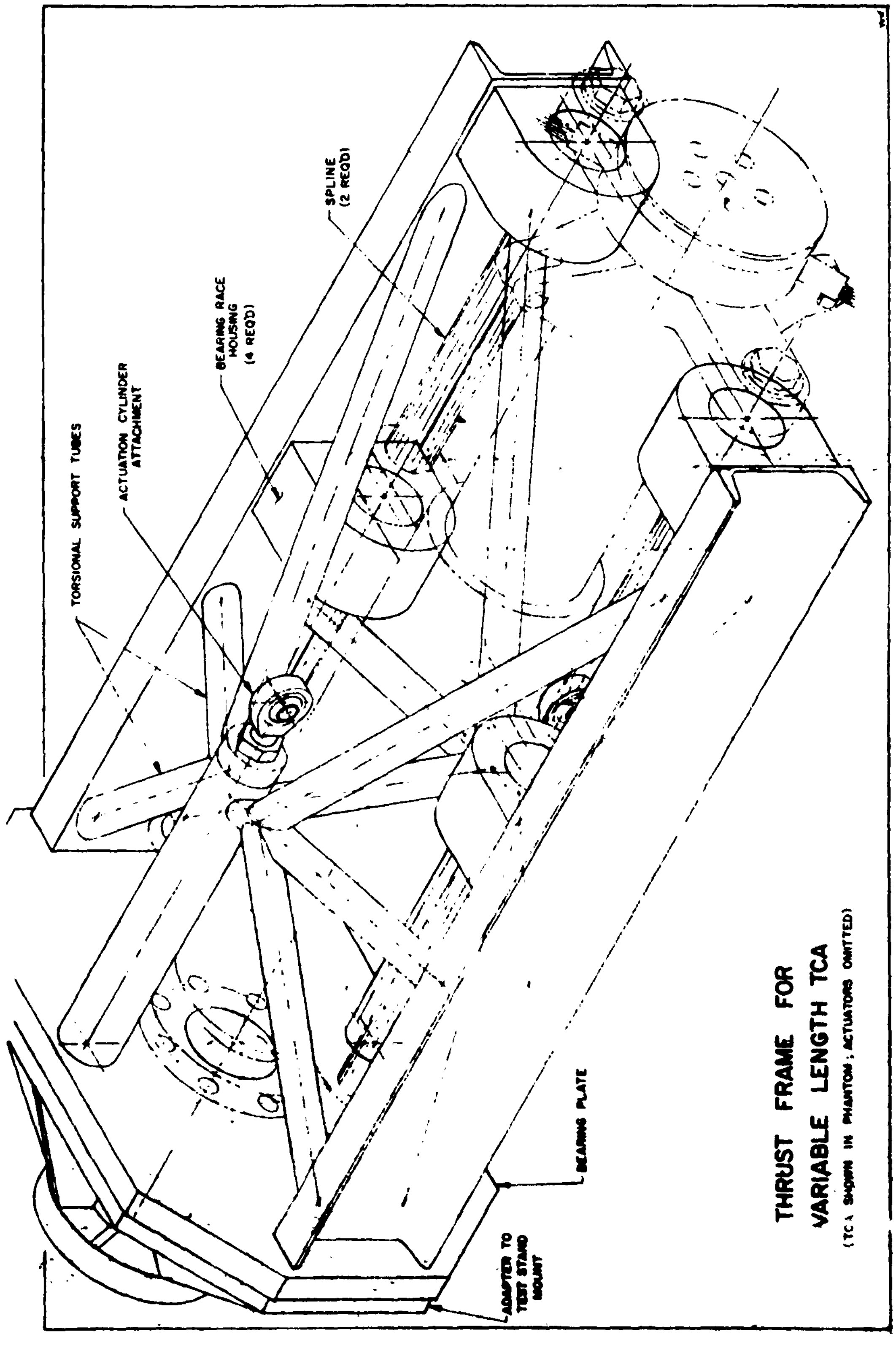
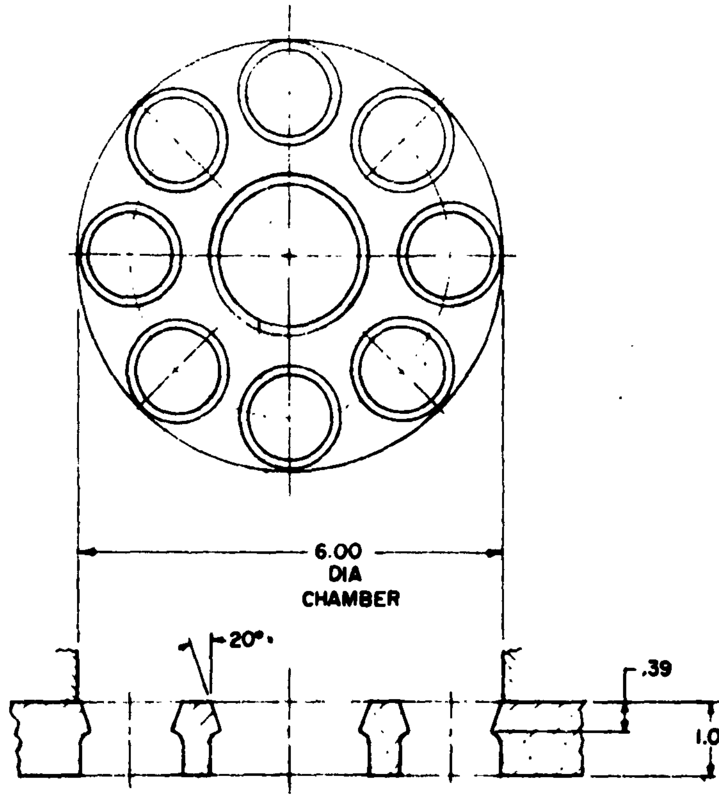

TORSIONAL SUPPORT TUBES
ACTUATION CYLINDER ATTACHMENT
BEARING RACE HOUSING (4 REQ'D)
SPLINE (2 REQ'D)
BEARING PLATE
ADAPTER TO TEST STAND MOUNT
THRUST FRAME FOR VARIABLE LENGTH TCA
(TCA SHOWN IN PHANTOM; ACTUATORS OMITTED)

Figure III-2



NOZZLE PLATE FOR VARIABLE LENGTH CHAMBER
TOTAL THROAT AREA = 11.66 IN^2.

9230-63-139

Figure III-3

IV. ABLATIVE THRUST CHAMBERS

A. INTRODUCTION

1. Purpose

The purpose of this project is to demonstrate the feasibility of large ablatively-cooled thrust chambers for high performance, liquid rocket engines operating at chamber pressures up to 300 psia for extended durations. The ablative chambers will embody the compression-molded building block concept.

2. Approaches

Six thrust chambers fabricated under Contract AF04(647)652/SA 33 will be used with injectors developed on the Apollo Service Module Engine Program (Contract NAS 9-150). A water-cooled transition will be used to adapt the Apollo injector to the ablative chambers which have a Titan II second-stage configuration. The thrust chambers are described in detail in the final report for Contract AF04(647)652/SA 33*, and are briefly described in Report 212/SA3-2.2-M-1 of the current series of monthly progress reports.

B. PROGRESS DURING THE REPORT PERIOD

1. Transition Section

The pacing item of the program is the water-cooled transition section which will allow the ablative chamber to be fired with an Apollo injector. For this reason, primary emphasis has been the completion of the heat transfer analysis and design of the transition section. The heat transfer analysis was completed one month ahead of schedule.

---

* BSD-TDR-63-118, "Ablative Thrust Chamber Feasibility, 28 June 1963.

A computer program was used to calculate heat transfer parameters, using a modified form of the Bartz correlation for gas-side film coefficients, the Dittus-Boelter correlation for non-boiling liquid-side coefficients, and the Gunther correlation for burnout heat fluxes. A factor of two was applied to the gas-side heat transfer correlation to account for combustion turbulence. For the basic design, the use of stainless steel tubes was assumed with coolant water available at 80 psia and 70°F. A design for a constant perimeter tube was selected in preference to a tapered tube for economy and speed of fabrication. A single pass coolant circuit was chosen over a multi-pass system to eliminate complexities in manifolds.

The heat transfer study resulted in a design incorporating 150 of 3/8-in. diameter tubes with an 0.020-in. wall thickness. The water flow velocity will be 35 fps at 190 lb/sec. The total pressure drop will be 3.2 psi, and the gas-side wall temperature will be 912°F. The maximum burnout heat flux ratio (ratio of actual heat flux to the burnout heat flux) attained in the transition section will be 0.57.

Final design of the transition section was completed ahead of schedule at the end of July. Fabrication will be initiated immediately.

2. Injector

An injector without baffles has recently been stably test fired in the Apollo Service Module Engine Program. Additional firings with this injector and others of identical design will be made on that program for higher confidence in the stability of performance of this design. Because of the anticipated high demand for this type of injector in the Apollo development,

it is not yet certain when an injector might be available for use in the Product Engineering Program. The fabrication cost for this injector is presently estimated to be in excess of that allocated in the Product Engineering Program for that purpose. An analysis will also be required to determine whether the Apollo aluminum injector is capable of withstanding the higher chamber pressure required (300 psia vs 100 psia) or the associated higher heat fluxes without burning out the face.

3. Ablative Chambers

Two additional steel shells are being fabricated and minor changes are being made in the interior contour of the ablative sections at the forward end to adapt to the transition section. As the new steel shells are completed, the bonded ablative sections will be inserted for machining of the interior contour.

C. NEXT REPORT PERIOD

Major emphasis will be placed upon fabrication of the transition section. The two new steel shells will be completed and ready for installation of the ablative sections. Close coordination will be maintained with the Apollo Service Module Program on their progress in injector testing.

TABLE IV-1

ABLATIVE THRUST CHAMBERS FEASIBILITY MILEPOST CHART

CURRENT CHANGE	ABLATIVE THRUST CHAMBER FEASIBILITY	PRESENT SCHEDULE			PRIOR	Calendar Year 1963							Calendar Year 1964				
		DAY	MO	YR		J	J	A	S	O	N	D	J	F	M	A	M
	PHASE I, TEST PREPARATION																
	A. Design of Ablative Chamber				A												
	B. Fabrication of Ablative Chamber																
	1. Chamber 2-452[illegible]				A												
	2. Chamber 27[illegible]120-14[illegible]	31	8	63				◆									
	3. Chamber 27[illegible]120-[illegible]	31	8	63				◆									
	4 Chamber 27[illegible]120-1	31	8	63				◆									
	5 Chamber 2[illegible]452[illegible]-2[illegible]	31	8	63				◆									
	6. Chamber 27[illegible]120-[illegible]	31	[illegible]	63				◆									
	C. Design of Metal Shell				A												
	D. Fabrication of Metal Shell	31	[illegible]	63				◆									
	E. Design of Transition Section	30		63				A	◇								
	1. Heat Transfer Analysis	31		63			A	◇									
	F. Fabrication of Transition Section	30	10	63						◆							
	G. Selection of Injectors	31		63				◆									
	H. Injector Fabrication	30		63					◆								
	I. U[illegible]lel Chamber Acquisition					A											
	J Selection of Instrumentation	30		63					◆								
	K. Selection of Firing Parameters	15	11	63							◆						
	L. Ablative Chamber Assembly	31	12	63								◆					
	M. Installation of Instrumentation	30	11	63							◆						
	PHASE II, FIRING TESTS																
	A. Injector Checkout	30	11	63							◆						
	B. Ablative Chamber Test	31	1	64									◆				
	PHASE III, DATA EVALUATION																
	A. [illegible]	21	2	64										◆			
	B. [illegible]																
	[illegible] rmance	[illegible]	3	64											◆		
	[illegible] Evaluation	1[illegible]	3	64											◆		
	[illegible] With Subscale Tests	15	4	64												◆	

V. PROGRAM REPORTING

Table V-1 is the schedule for contract reports.

TABLE V-1

PROGRAM REPORTING MILEPOST CHART

CURRENT CHANGE	PROGRAM REPORTING	PRESENT SCHEDULE DAY	MO	YR	PRIOR	1963 J	J	A	S	O	N	D	1964 J	F	M	A	M	J
	1 FINANCIAL STATUS REPORT																	
	1st Quarter	[illegible]	[illegible]	63														
	[illegible] Quarter	[illegible]	1[illegible]	[illegible]														
	3rd Quarter	[illegible]	3	64														
	4th Quarter	[illegible]		64														
	2. FUNDING STATUS REPORT																	
	June 1963	10	7	[illegible]3			A											
	July [illegible]	10		[illegible]														
	[illegible]	1[illegible]		[illegible]														
	[illegible]	1[illegible]	1[illegible]	[illegible]														
	[illegible]	[illegible]																
	[illegible]	[illegible]	[illegible]	[illegible]														
	[illegible]	[illegible]	[illegible]	[illegible]														
	January 1964	10	[illegible]	[illegible]4														
	February [illegible]	1[illegible]		[illegible]4														
	[illegible]	[illegible]		[illegible]4														
	April [illegible]	[illegible]	[illegible]	[illegible]4														
	May [illegible]	1[illegible]	[illegible]	[illegible]4														
	[illegible]																	
	J[illegible]	[illegible]	7	[illegible]			A											
	J[illegible]	[illegible]		[illegible]														
	[illegible]	[illegible]		[illegible]														
	[illegible]	1[illegible]	[illegible]	[illegible]														
	[illegible]	[illegible]	[illegible]	[illegible]														
	[illegible]	10	[illegible]	[illegible]														
	[illegible]	[illegible]	[illegible]	[illegible]4														
	[illegible]	1[illegible]		[illegible]4														
	[illegible]	[illegible]		[illegible]4														
	[illegible]	[illegible]	4	[illegible]4														
	[illegible]	[illegible]		[illegible]														
	[illegible]	1[illegible]	6	[illegible]4														
	[illegible] PROGRAM PROGRESS REPORT (MONTHLY)																	
	June [illegible]	[illegible]	[illegible]	[illegible]3			A											
	July [illegible]	10		[illegible]														
	September 1963	[illegible]	10	[illegible]3														
	October 1963	[illegible]	11	63														
	December [illegible]	[illegible]	[illegible]	[illegible]4														
	January 1964	20	[illegible]	[illegible]4														
	March 1964	[illegible]	4	[illegible]4														
	April 1964	[illegible]	[illegible]	[illegible]4														
	5. PROGRAM PROGRESS REPORT (QUARTERLY)																	
	August 1963	20		[illegible]3														
	November 1963	20	1[illegible]	[illegible]3														
	February 1964	20	3	63														
	6. FINAL PROGRAM REPORT	15	7	64														
	7. DOCUMENTARY FILM	15	7	64														

DISTRIBUTION LIST

External	No. of Copies
Commander Ballistic Systems Division Air Force Systems Command Norton Air Force Base, California Attn: BSRPA	1
Commander Ballistic Systems Division Air Force Systems Command Norton Air Force Base, California Attn: BSRPT	1
Defense Document Center Arlington Hall Station Arlington, Virginia	10
Space Technology Laboratories Norton Air Force Base, California Attn: Mr. A. I. Hoffman	1
Commander Rocket Propulsion Laboratory Edwards, California Attn: DGREA	5
Liquid Propellant Information Agency Applied Physics Laboratory John Hopkins University 8621 Georgia Avenue Silver Springs, Maryland	2
Internal	47